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Summary

Eros (433), the largest of the known close approach asteroids, will pass within 0.15 AU of the Earth during its 1975 opposition. This close approach, occurring near the asteroid's descending node and perihelion, offers an early opportunity for a relatively low energy manned interplanetary mission. Such a mission could provide data important to the space technologies, to the astrosciences, and to the utilization of extraterrestrial resources. Mars and Venus are the primary targets of early manned planetary flight, and therefore early manned planetary missions to other objects should support or complement the Mars and Venus programs. The model mission described in this paper satisfies this criterion and can be accomplished by a single uprated Saturn V and derivatives of MORL and Apollo.

The model mission includes a 527-day free return flight performed by a 330,000-pound vehicle system. An S-IVB/IU stage injects the spacecraft into a two-day geocentric ellipse where it is erected to its interplanetary configuration. The spacecraft departs with a second impulse and 270 days later passes within 50 miles of Eros where a turret-mounted instrument complex and an unmanned probe make a careful examination of the asteroid's surface. About 257 days later the reentry vehicle, a six-man Apollo-type command module separates from the spacecraft and returns the crew to the Earth's surface.

The mission is concluded to be technically feasible and an attractive complement to the manned interplanetary program.

Introduction

Manned flights to the near planets have been examined in considerable detail by numerous government and industry studies. On the other hand, manned missions to lesser bodies of the solar system other than the moon, such as the close approach asteroids and short period comets, have received relatively little attention. Except for recent NASA sponsored studies of unmanned systems, published material describing specific lesser body missions is sparse, and that related to manned flights is practically non-existent. In spite of their small size, the lesser members of the Solar family are important keys to unlocking the vast storehouse of solar system knowledge. Early missions to certain of these objects, in advance of manned missions to the inner planets, could be of significant value, not only in exercising Mars or Venus mission technology, but in obtaining data toward answers to fundamental questions in the astrosciences as well.

Some of the lesser bodies are more easily reached than the near planets, this fact suggesting the following questions:

> Are there lesser bodies, other than the moon, sufficiently interesting for an early manned mission?
>
> Are such missions technically and economically feasible?
>
> Can such missions complement, rather than compete with, the more ambitious Mars and Venus flights?

This paper presents a partial answer to these questions by examining the technical feasibility of a 1975 manned mission to the close approach asteroid Eros (433). The discussion is presented in three main parts. The first is a brief examination of the lesser bodies of the solar system and their accessibility to early missions. The second part describes Eros and mission alternatives for its study. The third part discusses the requirements for and implementation of a manned mission to this asteroid.

Part I - The Lesser Bodies: Characteristics and Missions

The solar family can be roughly divided into three groups of particulate matter: The Sun, the nine planets, and the lesser bodies. The latter group; containing the planetary satellites, asteroids, comets, meteoroids, and the interplanetary dust; are important interplanetary targets for three reasons:

(1) Any theory of origins must account for the wide variety and distribution of these objects and a close examination appears to be prerequisite.

(2) These bodies, especially the asteroids and satellites, are potentially valuable for exploitation. As the expanding space frontier places an ever increasing demand on terrestrial supplies, the utilization of the extraterrestrial resources stored in the lesser bodies will become not just a possibility, but an economic requirement.[1] An early determination of exactly what does exist in these bodies, and in what quantities, is a necessary step for future mission planning and implementation.

(3) The lesser bodies are important components of the interplanetary weather,[2] expecially as related to manned mission hazards. Flights beyond Mars, for example, will generally encounter the belt asteroids, Saturnian orbits must consider the hazards of the rings, and flights between the planets may require tailoring to avoid meteoroid and dust concentrations.

Characteristics of the Lesser Bodies

Planetary Satellites. The thirty-one known satellites circle six of the nine planets and range from 5 to

3550 miles in diameter. Seven of these, including the moon, have diameters exceeding 2000 miles. Predicted structures range from "stony" to low density "snowballs," and while some appear spherical, the light curves of others suggest more irregular shapes. The satellites appear to have a variety of origins; some are thought to have formed with the primary, others are thought to be captured asteroids or collision fragments. Some of them have particularly interesting features.[3] Jupiter's Europa, for example, has an exceptionally high albedo of ~ 0.75. Saturn's rings, technically a family of satellites, is one of the most outstanding features in the solar system. Another Saturnian satellite, Titan, is the only moon on which an atmosphere has been detected; and one hemisphere of Saturn's Iapetus is about five times brighter than the other, a tentative theory suggesting that part of one hemisphere was sheared flat by a collision.

Asteroids. The asteroids, or minor planets, move principally between the orbits of Mars and Jupiter with an average mean distance from the sun of ~ 2.8 AU. The largest asteroid, Ceres, was also the first discovered and since that time (January 1, 1801) the well determined orbits of over 1650 have been recorded, most of these being located in the main belt between 2.0 and 3.5 AU. Some non-belt asteroids pass inside the Earth's orbit and one, Icarus, passes between Mercury's orbit and the Sun. Hidalgo occupies a comet-like orbit inclined over 42° and moves outward nearly as far as Saturn. The Trojan asteroids orbit near Jupiter's L4 and L5 libration centers, and the many undiscovered asteroids which possibly lie beyond the belt may extend the minor planet population to the boundaries of the solar system. Measured diameters have been obtained for only a few of these bodies with values ranging from about 480 miles downward, most sizes being estimated from magnitude/diameter relationships. The asteroids are studied mainly by reflected light and most are thought to have lunar-like albedos. Similarities with the moon include color, brightness-phase relations, and polarization characteristics.[4] Lightcurves, brightness vs. time, indicate rotation rates of two to twenty hours, both direct and retrograde; the rotational axes roughly aligned and the poles inclined about 30° to the ecliptic.[4] Average asteroid density is probably about 3.5 gm/cm^4; however, neither mass nor density has been determined for any orbiting asteroid.

Comets. These bodies appear as bright objects, usually with a tail, moving in nearly circular to nearly parabolic solar orbits. Observed by reflected sunlight or by induced fluorescence, comets consist of a head containing an apparently solid mass surrounded by a cloud ($\sim 10^5$ mi.), the outer parts of which blend into the tail. The nucleus is thought to hold the key to understanding the physical nature of comets[5] although some show no such well defined area. A currently accepted comet model is Whipple's "icy conglomerate" which consists of a nucleus made up of frozen gases containing bits of solid meteoric particles; as the comet approaches the Sun, sublimated material provides the gas and particles making up the coma and tail. Comet sightings have been reported throughout recorded history, and apparations of more prominent comets can be traced far into the past; but in spite of their long history of observation, relatively little is known about comets beyond that derived from their motion and light.

Other Lesser Bodies. The meteoroids, interplanetary dust, and other bits of small particulate matter generally prevail throughout the interplanetary space. Much of this material is probably the result of collisions between asteroids, comets, satellites, and planets; from the debris of degenerating comets; and perhaps from planetary ejections. Some may be the result of a continuing process of accretion; certainly some is swept from interstellar space, and some contributed by the Sun. These objects range in size from asteroidal to dust, the exact divisions not well defined, and often travel in "clouds" with cometary orbits. The very small bodies tend to be more uniformly distributed, although concentrations are associated with the meteor streams, regions such as the lunar libration centers (the Kordylewski Clouds), and in the vicinity of the planets. However, this finer material generally tends to move in nearly circular orbits, gradually spiraling into the Sun.

Manned Missions to the Lesser Bodies

The 1970's will probably mark the beginning of manned interplanetary flight. Unmanned spacecraft will continue to be the workhorses of the time period, however, with potential targets among all known members of the solar family; they will be the trail blazers, probing the new environments and laying the groundwork for the more ambitious missions to follow.

Lunar Class Missions. The best known lesser body manned missions are those currently being implemented in the Apollo lunar program. Missions to the Kordylewski Clouds at the lunar libration centers are probably no more difficult and entirely within the capability of the present Apollo lunar system; in fact, such a flight might be an attractive AAP mission, using a Pallet payload to examine or collect specimens of the interplanetary population.[6]

Interplanetary Class Missions. The economic and technological constraints of the 1970's appear to preclude manned missions to the known satellites of any planets except Earth and Mars. Further, it does not seem reasonable to expect manned missions to specific concentrations of meteoroids, partly due to the hazard involved; nor to concentrations of micrometeoroids and dust. Nor do manned missions to the belt asteroids appear likely, at least in the early part of this time period; however, in the later 1970's, flyby missions to the Mars side of the belt may be a feasible part of a dual mission including Mars or Venus or both--provided the particulate environment is defined by earlier unmanned probes. Manned missions deep into the belt appear even less likely, and flights beyond do not appear probable until the 1980's or later.

Among the interplanetary missions which might be considered in this time period are the Mars and Venus flybys. While these can provide significant returns for the planetary sciences as well as astronautics in general, the development of orbital assembly or new launch vehicle upper stages, possibly nuclear, is probably a requirement. Missions to certain close approach asteroids and short period comets, on the other hand, are significantly less demanding. Velocity requirements from a low Earth orbit for manned flybys to some short period comets and close approach asteroids are in the range of 13,000 - 15,000 ft/sec, and appear to be compatible with single launch missions using uprated Saturn V class launch vehicles[7]. The comets provide larger targets for intercept and flyby missions than do the asteroids, but tend to have considerably higher relative velocities at encounter. Comet orbits are generally less certain than those of the close approach asteroids and the allowable launch windows are therefore more difficult to establish; careful tracking prior to launch is required. The comets are more amenable to early sampling missions than the asteroids, but being accompanied by considerable particulate matter are likely to impose an unacceptably high risk on a manned mission, especially for a flythrough.

Close Approach Asteroids: Target of the 1970's. The close approach asteroids are among the most attractive interplanetary targets for manned missions of the next decade[8]. Not only can such missions provide significant new astrophysical data, but the manned flights can simultaneously acquire invaluable spaceflight experience directly applicable to the more difficult planetary missions, and can accomplish this without orbital assembly, without new upper stages, and with the hardware and technology being developed and man-rated for programs currently underway. Mars and Venus missions enjoy a priority at present but require a higher order of technology (orbital-assembly and/or nuclear or higher energy chemical upper stages). Early cometary missions impose intercept requirements probably better suited to unmanned probes. Missions to planetary satellites, except those of Earth and Mars, and most belt asteroids, appear too ambitious for manned flight in the 1970-1980 period; and the concentrations of meteoroids and dust, with the possible exception of the Kordylewsky clouds, are, like the comets, probably better suited to unmanned probes.

Part II - Eros

Selection. Of the known close approach asteriods, Eros is a particularly good choice for an early mission, either manned or unmanned, for several reasons:

(a) the degree of precision with which the orbital characteristics are known is high, largely due to the use of Eros in astrometry.

(b) Eros is the largest of the known close approach asteroids, making mass determination by flyby less difficult than for other known close approach bodies.

(c) The asteroid moves in a readily accessible orbit.

(d) Eros is about the same size as Phobos and Deimos, facilitating a direct comparison with the moons of Mars.

(e) Eros is probably a relatively recent object originating as a collision fragment and possibly showing "cross-sectional" features of a larger parent body and possibly showing structural deformation from an impact.

History. By 1898 the number of known asteroids had increased to over 400. The general uniformity in appearance and orbital characteristics led many astronomers to question the value of time spent in searching for these objects and in defining their orbits. Then, on August 13, 1898, G. Witt photographically discovered an asteroid[9] of the 11th magnitude retrograding at the daily rate of half a degree. This unusual motion caused it to be widely observed, and initial elements for the new asteroid, identified as 1898DQ, were soon published. After a subsequent refinement of its orbit, Eros was given its permanent number: 433. In 1900, von Oppolzer observed Eros to be changing greatly in brightness; the cause of the variation, now attributed with reasonable certainty to a rotating irregular shape, was originally suggested for earlier discoveries by Olbers in 1802[10]. This view was confirmed during the 1931 close approach when Eros was telescopically observed to be "brick-shaped" with a direct rotation and to be about 22 km long[11].

Eros is probably best known for the part it played in determining, or refining, the values the solar parallax, the lunar inequalities, and the mass of the Earth-Moon system.[12] Of three independent methods for determining the solar parallax, two involve the use of a close approach asteroid, and Eros was used for this purpose by several investigators from 1901-1945. The ratio of the Moon's mass to that of the Earth, a ratio of fundamental importance to the determination of astronomical constants, can be obtained from measurements of the "lunar inequality" produced by the motion of the Earth's center about the Earth-Moon center of mass. Eros was used as the reference body for this measurement in 1909 and again in 1941 and 1950. Precise determinations of the mass of the Earth were made in 1921, 1933, and in 1940 by measuring the perturbations of Eros caused by the Earth-Moon system. In addition, Eros has been used for determining the mass of other planets such as Mercury, Venus and Mars. An important result of this activity is that the orbital characteristics of the asteroid[13] (Figure 1) are known to a relatively high degree of precision.

Description. Eros is most generally described as a "brick-shaped" or elongated body with an irregular surface (Figure 2). The physical characteristics of this asteroid, however, like all other asteroids, are relatively unknown. Is it a fragment from some larger body or is it an original formation? What is its mass? What surface features does it show? Does the asteroid hold evidences of extraterrestrial life? Does it contain substance of important economic or strategic value? Is there a correlation between the Martian moons and this asteroid which about the same size as Phobos? Answers to such questions are within the reach of our present technical capabilities plus the predictable technologies of the immediate future.

Manned Missions to Eros.

By 1975, a manned mission to Eros could, in addition to satisfying scientific objectives, provide invaluable experience for the more difficult flights to Venus and Mars. An unmanned mission would be adequate for obtaining valuable asteroidal information since almost any data obtained by an intercepting space probe would represent a "quantum jump" in minor planet knowledge. But, the opportunity for orders of magnitude increases in the quality and quantity of this potential data through a manned mission is made doubly attractive by the added contribution of the experience and training to the later Mars and Venus flights. The present of trained scientists-astronauts in the spacecraft would permit "on-the-spot" selection and examination of previously unknown features of immediate importance and real time evaluation of unknowns. Further, the ability of the crew to navigate provides for the improved on-board tracking and course correction necessary to insure a small encounter miss distance. Another consideration is the classic advantage of manned missions: the enhanced mission success available through an on-board capability for maintenance and repair. And finally, the value of the Eros mission to subsequent manned planetary flights having a higher level of difficulty and complexity is of no small consequence. While much of the experience needed for interplanetary flight will come from simulation and from Earth orbital and lunar missions, interplanetary experience comes only from interplanetary missions; less difficult flights, such as that to Eros, could significantly enhance experience acquired in Earth orbital and lunar activities, and could thereby increase the probability of success for the missions to follow.

Flyby missions are attractive early mission alternatives due to their low energy requirements, but they permit only short stay times in the vicinity of the asteroid. Course deflections due to a close approach are nearly insignificant as indicated in Figure 3, -- unless the density of Eros is significantly larger than the lunar-terrestrial density range shown.

Impact missions are also attractive. A secondary vehicle, or spacecraft dispensed particles,[14] impacting on Eros could be used to obtain details of surface characteristics and perhaps asteroidal mass, although the latter may be particularly difficult.

Rendezvous and/or orbit at Eros would greatly increase the value of the mission and would permit a much more detailed study of the surface features. In addition, a secondary vehicle could provide a fairly good estimate of the asteroid's mass by measuring acceleration as it approached the surface. A close position, or orbit, by either the primary or a secondary vehicle might also indicate variations in the density of Eros. A rendezvous or orbiting mission, however, is a relatively high energy event.

A landing on Eros, either manned or unmanned, is probably the most attractive mission alternative, and also the most expensive. The landing would, of course, be performed by a secondary vehicle; the problems associated with landing even a small manned interplanetary vehicle on bodies such as Eros make the use of secondary vehicles a clear choice. The low surface gravity, Figure 4, and the low escape velocity, Figure 5, permit the use of small thrust chambers and low propellant fractions in these secondary vehicle systems. Man's ability to "walk" on the surface, Figure 6, may be restricted by this low gravity,[15] particularly near the ends of the elongated planetoid where the effects of centrifugal relief and surface velocity are the most pronounced.

An intercept at or near perihelion in 1975 occurs during a close approach period where communications distance is short; in fact, the free return trajectories are such that vehicles are never very far from Earth orbit and remain essentially in the ecliptic plane. Intercept at or near aphelion places the vehicle at about 1.78 AU, beyond the orbit of Mars; such an intercept in this time period would occur in late 1975 or early 1976 and involve communications distances in excess of 1.5 to 2.0 AU. Intercept at significant distances from the nodes involves a considerable plane change ΔV increment due to the inclination of Eros' orbit. These latter alternatives were not examined during the brief study reported here since they appear to require an initial mass in Earth orbit of about the same magnitude as that for a Mars or Venus mission.

The alternative that appears most suitable for an early manned flight is the flyby at perihelion, with the spacecraft carrying a secondary vehicle to impact the asteroid. Preliminary studies indicate the possibility of such a mission during the 1975 close approach using a single Saturn V and the systems and techniques from Apollo and its contemporaries; such a mission is described in Part III.

Part III Mission Requirements and Implementation

Feasibility Criteria and Approach

The technical feasibility of the 1975 Eros flyby mission was assumed to be established if a practical baseline spacecraft could be defined that was consistent with confidently predictable technology of the 1970-1975 time period, and if the baseline spacecraft could effectively utilize systems, equipment, and techniques being developed for Apollo and its contemporaries. The latter criterion reflects considerations of both cost and schedules. The approach to establishing this baseline was as follows:

(1) Establish a set of requirements and guidelines compatible with the 1970-1975 time period.

(2) Define a Model Mission consistent with crew safety, mission success, and state of art.

(3) Define a family of baseline spacecraft and identify "best-bet" design points for each.

(4) Compare the design points and rate for feasibility by judgment.

Requirements and Guidelines

The principle requirements, assumption, and guidelines used for this study are briefly discussed below.

State of Art. An underlying requirement for this study was the use of hardware and technology presently being developed for Apollo and its contemporaries, and where the characteristics of early time period equipment and techniques were efficiently applicable, such as used. However, study constraints which do not provide for growth, uprating, and new developments are unrealistic, and therefore, when predictable improvements were indicated by current development trends, conservatively extropolated characteristics were selected, but restricted to the underlying constraints of the Apollo period.

Crew Considerations. The mission is assumed to require a six-man crew, each crew member having a prime group of skills but capable of performing alternate tasks to provide a minimum of 100% redundancy for critical functions. Each crew member is assumed to be trained in general mission functions such as navigation, routine maintenance, and vehicle control; and each function covered by a specialist charged with ensuring a general proficiency in that function throughout the mission. The crew would generally operate in two-man teams, but during such events as trans-Eros injection, midcourse correction, the critical part of the encounter, and re-entry, the full six-man complement would participate.

Artificial Gravity. The need for zero g compensation is assumed to have been established by Earth orbital operations, and the spacecraft required to provide this. Of the two presently considered techniques, periodic centrifugation and spacecraft rotation, the former is assumed to be satisfactory. This is not to imply a clear-cut choice, however; centrifugation was chosen because (a) it imposes fewer configuration problems on the spacecraft, (b) eliminates vehicle spin-up and despin requirements, (c) facilitates continuous tracking of Earth, Eros, the Sun, etc., and (d) facilitates the orientation of critical heat rejection or solar cell surfaces.

Orbital Assembly. The spacecraft was designed for a single-launch mission; the use of multiple launches with Earth orbital rendezvous, an attractive but more costly alternative, was only considered as a backup if the spacecraft weights become excessive or if uprating the Saturn V does not prove adequate. In the latter event, an attractive alternative is the use of an SAT-V/2x156BA (Saturn V class)[16] booster to launch a 289,460 pound Mission Module and departure stage, and an S-1B/Stage O (Saturn 1B class)[16] to launch an 80,000 pound Service/Command Module combination. This approach facilitates nearly simultaneous launches and allows a 39,000 pound budget for rendezvous and station keeping and 330,000 pounds for the spacecraft.

Launch Vehicles. The launch vehicles assumed for this study are the uprated Saturn V class vehicles which have 100 n.mi. orbital payloads in the 300,000 to 400,000 pound range, achieving this performance with uprated or advanced versions of the Apollo Saturn V engines,[17] by increased propellant tank volumes[17] and higher energy propellants, or by additional stages such as strap-on solids.[16] The latter alternative appears to be particularly attractive due to the height restrictions imposed by the Vertical Assembly Building, the Launch Umbilical Tower, and the Mobile Service Structure. The S-IVB stage is used together with the IU for two functions; to provide the final increment of the boost to parking orbit and the main injection impulse into the trans-Eros trajectory. Uprated versions of the S-IVB have been described[17]; the version used in this study, however, is assumed to be an essentially unmodified Apollo version except for an increase in structural strength due to the larger payload weight and size. The instrument unit is also assumed to be similar to that used for Apollo with the exception of increased structural strength. New high energy upper stages for use in place of the S-IVB may be available in the time period of interest to increase the orbital payload or reduce the overall height of the launch vehicle, or both; such stages could also be the outgrowth of interplanetary or lunar injection requirements. However, these alternatives were not included in this study.

Support Facilities. Major ground facilities, such as those required for assembly, checkout, launch, tracking, and command, are assumed to be available or adaptable from other programs. To avoid costly and time-consuming alterations, it is expected that certain reasonable constraints will be imposed by these facilities on the Eros mission; for example, the overall height of the system may be limited by clearance in the Verticle Assembly Building for assembling and removing the space vehicle. Orbital facilities one may expect for supporting the mission include relay satellites; and optical tracking from Earth orbit or the Moon may supplement ground based RF systems. An orbital, and perhaps cislunar or lunar, rescue capability can be expected in the mid-1970's and the Eros vehicle departure maneuver was tailored to make use of this possibility.

Model Mission

Trajectory. The model mission is based on a free return trajectory developed by Dr. R. Dunn and depicted in Figure 7. The trajectory has several features important to this early manned planetary flight. First, the low energy is compatible with a single launch; the considerations related to orbital assembly, e.g., multiple-launch, rendezvous, the coupling and checkout of large modules, etc., are not required. Another feature is the short communications distance; the spacecraft is always within about thirty million miles of the Earth and during the critical encounter period when the data rate is the highest this distance reduces to about fourteen million miles. Further, the low inclination of the transfer orbit to the ecliptic eases the navigation task, and finally the low energy flight includes a low unbraked re-entry velocity of about 38,000 ft/sec.

Mission Profile. The model mission profile is similar in several respects to the Apollo lunar mission and is shown in Figure 8. An important feature of the profile is the eliptical departure orbit in which the spacecraft is erected to the interplanetary configuration after receiving over 90% of its departure velocity; either departure or abort-and-earth-return is readily accomplished from this orbit and the spacecraft is readily accessible for rescue should this latter eventuality be required.

Launch Window. The thirty-day launch window depicted in Figure 9 establishes most of the spacecraft's onboard velocity budget. The ΔV requirements in excess of 12141 ft/sec (the nominal) include increments for both departure and a post encounter trim maneuvers.

Trans-Eros Injection. Trans-Eros injection is performed by a two-impulse maneuver: an initial large impulse by the S-IVB and a second smaller impulse by the spacecraft's Service Module engines. The magnitude of each impulse was determined by comparing the two tradeoffs shown in Figure 10; (1) the initial weight in Earth orbit vs spacecraft ΔV increment, and (2) the period of the eliptical departure orbit as a Function of the ΔV increment provided by the S-IVB. Minimum initial weight in Earth orbit occurs with a spacecraft ΔV increment of about 3000 feet per second. The curve is relatively flat in this region and non-minimum values ranging from 2000 to 4000 ft/sec can be considered with only a small penalty. The departure orbit period, on the other hand, is quite sensitive to increasing S-IVB ΔV increments which rapidly lead to unstable orbits and unacceptably long interimpulse delays. An S-IVB ΔV increment of 9700 ft/sec was selected for this departure maneuver as shown by the design point on Figure 10.

The Encounter. The encounter period is defined as the two-day period centered around the point of closest approach. During the encounter period the velocities of Eros and the spacecraft are essentially constant and the asteroid and spacecraft move in approximately straight lines. Miss distances were assumed to range from 25 to 100 miles with position uncertainty of ±10 miles normal to the flight path and ±20 miles along the flight path. The spacecraft passes on the day side of the asteroid to assure that Eros is well illuminated by the Sun during the close approach interval, that the collision danger is minimized, that Earth-spacecraft communications will not be obstructed by asteroid, and that the instrument line of sight (LOS) can be maintained throughout the encounter with a constant spacecraft attitude.

Miss distance selection involves a trade between (1) guidance accuracy requirements and LOS rates, large allowable miss distances being an advantage for each, and (2) observation of asteroidal surface features for which small miss distances are desired. Nominal values much smaller than 25 miles are accompanied by high LOS rates, stringent guidance accuracies, and increased collision probabilities. Miss distances in excess of 100 miles are probably too large for effective resolution of surface details with reasonable spacecraft-mounted equipment. As shown in Figure 11, a value of 50 miles was chosen to provide a reasonably good balance between the variables of LOS rate and r. A more rigorous selection will require consideration of specific guidance accuracies and sensor turret and turret-mounted instrument characteristics. The distance between the spacecraft and Eros near the midpoint of the encounter period is also shown in Figure 11. Eros closes at an essentially linear rate until within about 30 seconds of encounter point. The spacecraft is within 200 miles of the asteroid for a period of about 90 seconds and within 100 miles for a period of about 30 seconds. The need for high speed data acquisition is apparent.

The Spacecraft

The Concept. The Eros Flyby Space Vehicle consists of an uprated Saturn V launch vehicle and the Spacecraft. The Spacecraft consists of (1) a Trans-Eros Injection System (an S-IVB stage and Instrument Unit as previously noted); and (2) the Basic Spacecraft which is composed of an Eros Command Module (EMC), and Eros Service Module (ESM), and Eros Mission Module (EMM). The EMM is non-propulsive and in many respects resembles MORL designs. Two versions of primary power, nuclear and solar, were considered. The ECM serves the same general function as the Apollo command module, and is examined here for two cases, ballistic and lifting re-entry. The ESM provides the spacecraft's propulsion and attitude stabilization requirements, and emergency power from fuel cells used prior to activation of the primary power source. The in-transit configuration consists only of the basic spacecraft; the S-IVB and IU being jettisoned after repositioning.

The four versions of the spacecraft are identified as follows:

(a) Concept I; Ballistic Re-entry and Solar Power

(b) Concept II; Ballistic Re-entry and Nuclear Power

(c) Concept III; Lifting Re-entry and Solar Power

(d) Concept IV; Lifting Re-entry and Nuclear Power

Concept I was selected as the most promising alternative, the intransit configuration being illustrated in figure 12. The prelaunch configuration is compared with the Apollo Lunar spacecraft in figure 13 to illustrate an important characteristic - the relatively short length of the Eros vehicle. Principle elements of the basic spacecraft are identified in figure 14.

Eros Mission Module. The Eros Mission Module (EMM) provides the main crew space and most of the non-propulsive subsystems for the flight. The scientific instruments are housed in this module together with the onboard equipment for processing, analyzing and storing the acquired data. Equipment maintenance and repair facilities and the main spare parts contingent are also located here. The EMM is envisioned to be similar to MORL concepts, and the Earth-orbit configuration of the MORL could almost be used for the flight without modification[17]. An "on-board emergency survival" design philosophy is incorporated since mission abort/escape possibilities appear to be limited to the early part of the flight.

The six-man 5200 cubic foot crew cabin complex consists of three sealable, airlock connected compartments, one of which is located in the Eros Command Module, the other two being located in the EMM. The largest of the three, a 4500 cubic foot compartment contains the crew quarters and the majority of mission work stations. This compartment is divided into three parts; a mission task area, a centrifugation area, and the crew quarters. The crew quarter's area is further divided into three 2-man cubicals and a lavatory-hygiene cubical. Each of the crew cubicals can be individually isolated if the need for a sick bay arises. A fifth division, the stormcellar/airlock, provides solar flare protection and an airlock connection to the ECM. The centrifuge area is defined by the swept volume of centrifuge. The mission task area contains the gally, the data processing and analysis equipment, the primary mission control station, and lab. A 400 cubic foot sealable equipment compartment, airlock connected to the task area, contains recessed panel mounted equipment racks for the remote electrical and electronic equipment and the spacecraft shops for servicing and repair. The main power distribution and conversion equipment is located in this compartment which can be isolated or rapidly decompressed in an emergency.

Life Support and Environmental Control[18] are provided by an integrated self-contained system.

(a) The semi-passive thermal-conditioning subsystem is sized to reject up to 15 Kw of heat from any two of four 350-ft^2 radiator complexes located in the outer EMM skin. The main coolant loop runs from the radiators to the low temperature equipment heat exchangers and the water condensing apparatus, then proceeds from the higher temperature equipment through an equipment-heating heat exchanger, and back to the radiator. Separate loops condition the equipment compartment to facilitate emergency sealing and depressurization.

(b) The atmosphere supply system uses a Sabatier reaction for oxygen recovery. Additional makeup oxygen can be provided from supplies in the water-management system; however, the primary source of makeup 0_2 is the storage tanks which also provide for cabin leakage and repressurization. The assumed leakage rate of 10 lb/day for the total spacecraft is the largest increment of the storage requirements, and assuming an overall oxygen recovery efficiency of 90%, the total requirement is 11.08 lb/day for the six man crew. Six cabin repressurizations are assumed for the total spacecraft. Atmospheric conditioning, i.e., CO_2 removal and contaminant control, is assumed to use a regenerable solid absorbent and catalytic burner although other systems such as molecular seives appear to be competitive.

(c) The water management system combines the reclamation of atmospheric water, wash water, and waste water in a single system. Due to the metabolic water production, the crew's total water requirements can probably be satisfied by reclamation alone, even though the system functions at less than 100% reclamation efficiency. In this study, however, it is assumed that only 98% of the required water is reclaimed, the remainder being lost with the atmospheric leakage or stored with the waste as shielding. An additional 1000 pounds of water is carried as contingency, most of the water being stored as stormcellar shielding. The stored clean water can also be used by the atmospheric supply system as a source of oxygen or by the ESM fuel cells.

(d) The waste management system provides for the collection, treatment, and processing of crew produced waste. Particular concepts await empirical evaluation in MOL and MORL; however, for this study, it was assumed that a suitable system will be developed for collecting, treating, and automatic processing; and that the crew will only be required to remove the packaged waste to storage areas in the stormcellar.

(e) Cabin temperature control, ventilation, atmosphere filtering, and supply gas pressure regulation and mixing is provided by the cabin conditioning system. Suit loops are provided in each Eros Mission Module compartment and in each airlock as a backup system and for umbilical operation of suited crew members. Additional connections are provided for extravehicular activities and operations in the unpressurized areas. The cabin conditioning system also supplies coolant and heating to the food preparation center and is interconnected with the ECM to condition that module when it is docked during the intransit mode.

(f) Contaminant control is distributed principally among the systems for atmosphere supply and conditioning, water management, food management, and cabin conditioning, and relies on the human senses, particularly sight and smell, to supplement such instruments as a mass spectrometer and gas chromatograph.

(g) A diet of dehydrated foods, for both hot and cold meals, is stored as stormcellar shielding, the reusable containers being used by the waste disposal system. Crew furnishings, including such items as tables, seat/restraint, clothing, hygienic facilities, bunks, and personal storage lockers, are mainly distributed throughout the crew quarters area.

(h) The stormcellar airlock contains suit loop connections to permit shirtsleeve or spacesuit occupancy. About 15 gm/cm^2 of shielding[19] is provided by a combination of aluminum primary structure, borated polyethylene, and stored spare parts, food, water, and waste. The airlock is also equipped for limited Earth communication, for critical monitoring and control, and with limited life support for use during periods of extended occupancy.

(i) The crew centrifuge is equipped with two adjustable position cars and is electrically driven on a rail system circling the task area entrance to the stormcellar airlock. The cars are equipped for water ballasting to permit balanced use by one or any two crewmen.

The telecommunications system provides a two-way voice and data link between the spacecraft and GOSS, Eros probe tracking and control during the encounter, and intercommunication voice and closed circuit TV between various crew stations in the several modules. Communications with GOSS utilizes the Apollo unified S-band system supplemented with additional RF amplification and a 20-foot diameter erectable parabolic antenna. Eros probe tracking and control requirements are derived from the Gemini/Agena Rendezvous system, the tracking radars located in the EMM being duplicates of those in the Command Module. A combined Laser/RF experiment is integrated into the system to permit an evaluation in support of subsequent missions. The integration of the laser link is on a non-interference basis with the primary RF system and a total of 250 pounds[20] is allotted for this. The communications system is considered to be a critical system form the standpoint of crew safety and data recovery; the former in the event of a need for emergency advisory assistance from Earth, the latter in the event of failure to recover the contents of the ECM. The approach adopted in this study is to incorporate a complete Apollo (i.e., lunar) S-band system in the ESM/ECM, together with the primary mission communications control station, and to duplicate this system in the Mission Module which also contains the main antenna and the RF power amplifiers. This approach provides two completely redundant "short range" systems, complete with control station, and permits normal GOSS communications with Earth from either of two sealable compartments.

The on-board navigation and guidance of the Eros spacecraft is jointly performed by a monitored mechanized system and by independent backup observations and computations performed by the crew[21]. This is supplemented by Earth based tracking and computation. The mechanical system employs precision angular measurements made by the crew from ECM, the EMM space sextant and sensor turret star or planet tracker providing a backup. These measurements are referenced to mutually aligned 1MU's, one in the ECM and one in the EMM, and the data is processed by either of two computers, also distributed between the two modules. Controls and displays, including clocks, are located in both the ECM and EMM with the former being the primary station and the latter a secondary back-up. Manual navigation and guidance tasks performed by the crew[21, 22] involve the use of such items as compact plotting devices, navigation tables, and self-contained sextants. Regular use of these tools permits the astronauts to maintain a "running check" on the mechanical system, improve their navigation and guidance proficiency, and insure a smooth transition to a manual mode should the mechanized system fail.

The instrumentation and monitoring system consists of decentralized information collection centers which transmit time shared and continuous data to the main monitoring consoles in the Mission Task area. Selected data, especially that of a critical nature, is also displayed in the Eros Command Module.

The computing and data processing needs of the mission and spacecraft are provided for by a centralized installation in the Mission Module. A separate installation in the Command Module, for use when the ECM is detached, provides a backup for navigation and limited data processing.

Maintenance and repair provisions include tools, test and checkout equipment, spare parts, limited shop facilities, and suitable manuals. Most of the maintenance and repair functions are performed in the equipment compartment where test and checkout instruments are stored in recessed panel installations. Bench space, power sources, and other such needs are also included in this compartment which can be sealed off from the remainder of the life support enclosure to preclude the release of toxic materials into the main compartment, or charged with a high percentage of inert gas to reduce the possibility of fires. Small equipment items can be opened in gas filled "glove boxes"; however, larger items may require charging the entire compartment with inert gas and performing the maintenance or repair tasks in a spacesuit. Suit umbilical connections are provided for this eventuality. Maintenance and repair activities outside the pressurized compartments are supported by external umbilical connections providing both ECS and communication. Typical locations include the aft unpressurized area, the instrument turret, power modules, and communications antenna.

The scientific equipment in the Mission Module consists of a sensor turret, an unmanned probe system, numerous analytical tools and instruments, and spare parts and miscellaneous items.

(a) The sensor turret[23], figure 14, was conceived as a means for acquiring asteroidal data with directionally sensitive equipment during brief encounters in which the feasibility of slewing the entire spacecraft is questionable. Such encounters include single body events in which the line-of-sight rate is high and multiple body events such as may prevail on missions to regions densely populated with asteroids. An unmanned turret concept is used due to the size and weight constraints of the single launch spacecraft concept.

The turret, retracted into the spacecraft for launch and for major inflight maintenance, is balanced and driven against a momentum storage system to minimize perturbing attitude torques on and by the spacecraft. The turret provides both azimuth and elevation travel, with the elevation drive providing most of the movement during the encounter. The instrument envelope shown is sized to include a 5 foot reflecting telescope/camera with a 30 inch primary mirror in addition to radar, spectrophotometers, photometers, TV, etc. One primary objective of the instrumentation is to obtain data for calibrating photometric studies[4] performed on and near the Earth.

(b) The unmanned probe[24] is used to obtain close range data from Eros, and possibly data during impact. The probe, a 200 pound secondary vehicle, is launched from the spacecraft prior to, or during, the early phases of the encounter period and is controlled by the crew in the primary vehicle. Data is transmitted from the probe to the spacecraft where it is simultaneously stored and relayed to Earth. The probe is catapult launched about 8 hours prior to the point of closest approach, a velocity increment of 9.0 ft/sec being imparted at $\sim$.25 g's. It is expected that launch conditions can be determined with sufficient accuracy that the total error due to uncertainties in the positions and velocities of Eros and the spacecraft, together with the attitude and attitude rates of the spacecraft, will permit the non-maneuvering probe to be placed within 1 mile of the nominal impact position in space. The actual launch velocity will, therefore, be determined after the relative positions and velocities have been measured in the early part of the encounter period, permitting about 16 hours of tracking prior to launch. The instrument complex includes such instruments as a TV system, a magnetometer, photometers and spectrometers for surface feature and composition data, and instruments for mass measurements. The probe, equipped with a momentum storage attitude control system, is brought into the equipment compartment for maintenance and checkout, the catapult system being integrated with an onboard handling system to facilitate this.

(c) The analytical tools and instruments charged to the scientific payload are mainly located in the equipment compartment and in the task area of the main compartment where bench, panel, and storage space is allotted. Much of this equipment is integrated with other systems to reduce the spares inventory and to facilitate dual usage in other task areas or for other disciplines; the central computer and data processing system for example, supports the entire mission. Miscellaneous instruments for establishing environmental and other data are installed in various locations throughout the mission module.

The spacecraft power system consists of three integrated subsystems: a primary subsystem located in the Mission Module, an fuel cell secondary subsystem located in the ESM, and a rechargeable battery secondary subsystem consisting of battery packs distributed throughout the spacecraft. The 15 kwe primary system supplies a 9.5 kw average continous demand to the entire spacecraft, with intermittant peaks to 14 kw. The peak power demands during the encounter increase to 14.75 kw.

(a) The solar photovoltaic system used with Concepts I and III is shown in Figures 12 and 14. An output of 9.0 watts per square foot at 1.0 AU, including an 8% manufacturing degradation, was assumed achievable for the time period of interest, and a value of 1.80 lb/ft^2 for cells, structural substrate, inter-module wiring, and miscellaneous fittings and framing, was used as an average over entire panel area. A 10% degradation over a 500-day period was included and the delivered output was based on a $\pm 10^\circ$ solar alignment. Using these constraints, the power available at 1.1331 AU, including the total degradation (5% conservative) is about 6.21 watts per square foot. The required area, 2415 ft^2, is provided by eight twenty-foot disk panels. During the launch and injection phase, the panels are stowed in the base of the Mission Module, and deployed as shown in figure 12 after separation of the S-IVB/IU. During all propulsion events the panels are positioned and locked in the x-y plane for better structural loading. During the encounter the panels remain aligned with the sun; but due to the sensor turret alignment requirements the spacecraft x and z axes are rotated out by the ecliptic plane to assure an unrestricted line-of-sight to the asteroid; the solar panels are then placed in an intermediate position. While a solar photovoltaic system of the 15 kwe size has not yet been built and tested, the characteristics are well understood due to the large amount of accumulated laboratory and flight test experience. A solar photovoltaic system is an attractive choice for this time peroid from among the solar powered alternatives; orientation constraints, however, are a distinct general disadvantage of the solar oriented systems.

(b) The SNAP 8 power source used in Concepts II and IV is attractive for several reasons. For example, the SNAP 8 Mercury-Rankine system is more nearly developed than any other nuclear sources in the power range of interest. Further, solar orientation is not required, and finally, the SNAP 8 nuclear source is more representative of Mars and Venus class missions. Unattractive features include the radiation which requires heavy shielding and involves potentially severe hazards for in-flight maintenance. Development for manned mission rating is a necessity and the system requires a relatively large stowage volume. Because of shielding requirements, the reactor was located at about 100 feet from the crew stations, requiring a variable geometry structure to permit packaging the system into a reasonable vehicle length. Stowage and deployment is illustrated in figure 15. Two SNAP 8 concepts were considered; one employing dual reactors in a fully redundant installation,[25] and the other using a single man-rated reactor[26].

Most of the power conditioning, control, and distribution equipment is located in the Equipment Compartment. Power substations are located in the task area of the main compartment, in the stormcellar airlock near the ECM interface, and in the aft unpressurized area near the sensor turret. Circuit protection is provided at each substation for the branch circuits originating there, circuit protection at the central power station in the Equipment Compartment covers both branch circuits and substation transmission lines.

The primary structure consists of the outer shell and the pressurized enclosures. The outer shell transmits launch loads to the ECM/ESM; provides meteoroid and partial radiation shielding; supports the pressurized enclosures, instrument turret, power module and other such items; and contains the thermal control radiators. The pressurized enclosure complex is constructed of foam filled aluminum sandwich pressure walls with integral frames and stringers. Equipment loads in the large spherical compartment, and loads from the stormcellar/airlock are carried by the cubical dividers and floors to dual transverse rings which attach the pressure shell to the outer cylinder. The airlock interface at the ECM docking pad is equipped with a semiflexible load-relieving section that is installed after the ECM/ESM stabilizing structure is attached, this latter structure securing and aligning the ECM/ESM, and serving two additional functions; to transmit intransit maneuver and attitude control loads directly to the EMM from the ESM while maintaining a close structural alignment and to provide a "hanger" for protecting the ECM.

Eros Mission Module weight as a function of mission duration is illustrated in figure 16. This weight is distributed as shown in Table I.

SUBSYSTEM	NUCLEAR EMM	SOLAR EMM
Life Supt. & ECS	36,895	36,895
Avionics	2,000	2,000
Power	14,550	6,510
Scientific Equip.	3,000	3,000
Structure	13,500	11,600
Spare Parts	2,340	2,250
Miscl.	4,200	4,200
	76,485 lb	66,455 lb

TABLE I EMM WEIGHT AT 600 DAY DESIGN POINT

Eros Command Module. The ECM serves a function similar to that for the Apollo Lunar Mission: it provides for launch phase escape, injection abort, onboard mission control, and Earth re-entry. The ECM is sized for 10 days life support for the 6 man crew and a 40,000 ft/sec re-entry velocity. Two alternatives were examined for the Eros flyby mission; ballistic and lifting body modules.

(a) An Apollo ECM was used as the ballistic alternative. Numerous government and industry studies[19] have utilized six man versions of this concept for Earth-orbital, lunar, and interplanetary missions; the particular configuration used as a starting point in this study was developed for a Mars mission spacecraft[17]. Advantages of this approach include proven state-of-art, convenient installation, and relative light weight; disadvantages include the minimal six-man volume and the limited capability for landing site selection. A basic vehicle weight of 12,350 lbs at re-entry was assumed for the study; an additional 500 pounds for subsistance and 350 pounds for installation penalties increases gross weight to 13,200 pounds.

(b) The M-2 lifting re-entry vehicle was the second alternative examined. The six-man version was derived from the logistics vehicle[27] shown in Figure 17. Advantages of this type include the relatively soft re-entry, the convenient cabin geometry for internal arrangements, and control of the descent and landing. Disadvantages; less development than the Apollo type CM and more difficult to integrate into the Eros spacecraft, particularly for the Nuclear powered version. A basic vehicle re-entry weight of 13,000 pounds was indicated resulting in a 14,000 pound gross including the 500 pounds of subsistance and 500 pounds of installation penalty.

Eros Service Module. This module provides spacecraft propulsion, attitude control and spacecraft secondary power. The general arrangement illustrated in Figure 18, the version used with the Apollo ballistic shape, is the most likely alternative for an early mission.

(a) The propulsion system consists of dual RL10A-3 rocket motors in a Centaur-like installation; propellant is carried in dual LO_2 tanks and a single LH_2 tank. The RL10A-3 motors are gimballed for thrust vector control and to facilitate single engine performance. The tankage is sized to provide 4100 ft/sec to a 330,000 pound initial weight spacecraft, with a delivered I_{SP} of 435 seconds. A useable propellant capacity of 34,050 pounds is required; a total of 36,140 pounds is carried to provide for residuals, boil-off, losses and chill down.

(b) Attitude control is provided by a hybrid system incorporating control moment gyros (CMG) and reaction jets. The ACS supports the repositioning maneuver, abort, midcourse maneuver, and re-entry; and provides sun-line and encounter alignment. The long duration low level torques due to the solar "wind" appear to be the only significant first order disturbances and are readily countered by the CMG's; solar induced torques are most pronounced during the encounter when the spacecraft is broadside to the sun. Four self-contained reaction control jet modules, fully redundant and similar to those in the Apollo Service Module, are installed as shown in Figure 18. They provide the large torques necessary for repositioning and rapid spacecraft slewing, and provide the impulse necessary to desaturate the CMG's. The Control Moment Gyro system consists of four identical dual rotor single axis units, three of which are installed in the ESM (Figure 18) to provide ECM/ESM and spacecraft pitch and yaw and ECM/ESM roll; one is installed in the EMM as a spare and for spacecraft roll axis control. The CMG's are arranged in a fully redundant configuration[28].

The 8.0 kw ESM power system services the spacecraft during the erection and checkout period and provides spacecraft emergency power to backup the primary source in the EMM. The 2000 kwh system consists of dual 4.0 kw fuel cells systems, each containing two 2.o kw FCA's and a 20 kwh rechargeable battery. Two 80 ft^2 radiators located in the accessory compartment are time shared with the reactant conversion system.

The reactant conversion system utilizes the energy available from the 15 kw primary power source in the EMM when the spacecraft power demand is near the average continuous value of 9.5 kw. The system converts the water produced by the fuel cells into LO_2 and LH_2 and provides heat leak compensation for the cryogenic tankage. The system is sized to a 5.5 lb/day conversion capacity and requires 2.0 kw of power, this requirement reducing to about 250 watts for heat leak compensation only. Approximately 165 days are required for converting the erection phase water and therefore the full 2000 kwh emergency capacity of the fuel cell system is available prior to the encounter with Eros.

The ESM primary structure is generally like that in the EMM but more heavily insulated to reduce heat leaks into cryogenic tankage. Like the EMM, radiators are located in the external skin and arranged to reduce heat inputs to the propellant tanks. The propulsion LH_2 tank is supported by a conical, low conductivity structure; the LO_2 tanks by a system of struts. The propulsion tank bay is closed by a transverse bulkhead which joins the cruciform panel system supporting the equipment located in the accessory compartment.

ECM/ESM weight is distributed as shown in Table II for Configuration I and II; Configuration III and IV are slightly heavier due to the large spacecraft gross weight.

Element	Ballistic ECM	Lifting Body ECM
ECM	13,200	14,000
Propulsion (wet)	6,030	6,450
Attitude Control	2,925	2,980
Power	3,660	3,660
Reactant Conversion	820	820
Structure	4,750	5,870
Spares & Contingency	1,900	2,050
Total Wet Weight	33,285	35,830
Propellant	33,912	35,215
ECM/ESM Gross Weight	67,197 lb	71,045 lb

TABLE II, ECM/ESM WEIGHT

The Integrated Spacecraft. The four configurations formed by combining the various modules just discussed are compared in Figure 19 which shows the required initial weight in Earth orbit. The payload capability of a range of uprated Saturn V launch vehicles is also shown, this range mainly including versions with uprated engines, increased tank capacity, and strap-on solids. Although launch vehicles capable of boosting all four configurations are indicated, Configuration I - Solar Power and Ballistic ECM - is probably the better choice since it is closer to the present state-of-art and probably more compatible with the weight limitations of the Apollo ground equipment such as the LUT and the Crawler-Transporter. Configuration I and II also appear to be better choices relative to the height constraints imposed by the Apollo Facilities at KSC.

Mission Implementation

The following sequence of events illustrate the implementation of the nominal mission (Figure 7 and 8) with the Configuration I Spacecraft:

a. Boost to a 100 n. mi. Earth parking orbit by a boost augmented Saturn V. Perform pre-trans-Eros injection checkout, and inject into the elliptical departure orbit (ΔV=9700 ft/sec).

b. Perform repositioning maneuver, jettison the S-IVB/IU, and erect to the interplanetary flight configuration. Deploy ECM/ESM stabilizing structure and shielding, and complete interface connections with EMM. Deploy, activate and checkout primary power system. Transfer electrical loads from ESM fuel cells to EMM primary source and place fuel cells on standby. Activate and checkout all spacecraft subsystems: life support, communications, attitude control, regeneration, propulsion, etc. Deploy and checkout sensor turret; checkout all scientific instrumentation and equipment. Verify departure ellipse characteristics and determine 2nd injection impulse parameters. Verify GO status on all systems and prepare for 2nd injection impulse. If mission NO-GO at this point, abort.

c. $T = 0.0^d$: perform 2nd Trans-Eros injection firing with ESM (ΔV = 2441 ft/sec) and acquire Sun-oriented coast attitude (Figure 12).

d. Perform mid-course maneuvers. (ΔV budget = 300 ft/sec.)

e. $T + 210^d$: Rotate solar panels into x-y plane, lock, and re-acquire sun-orientation. Spacecraft now broadside to sun with x-axis parallel to ecliptic plane to facilitate tracking Eros, and Earth communications.

f. $T + 260^d$: Begin Encounter Tracking. $T + 265^d$: Perform Encounter Trim Maneuver, ΔV budget = 75 ft/sec. $T + 265^d$: Orient Spacecraft for the Encounter; position z-axis parallel to the predicted LOS at the encounter point and roll about z-axis to place the Sun-oriented solar panels below the trajectory plane with the vehicle x-axis normal to the asteroids flight path within $\pm 5^\circ$. Resume Encounter tracking and initiate Encounter data acquisition. Encounter minus 8.1 Hours: Launch probe. $T = 270^d$: Encounter point (closest approach); maximum data acquisition rate from on-board sensors and from probe. $T + 272^d$: Terminate Encounter Tracking. Perform Return Trajectory Maneuver, correct for Encounter Trim Maneuver, etc. (ΔV budget = 75 ft/sec + part of balance from 2nd injection firing). Acquire Sun-oriented altitude.

g. $T + 330^d$: Unlock solar panels and rotate to position normal to x-axis. Reacquire Sun-orientation.

h. Perform Mid-course Maneuvers. (ΔV budget = 300 ft/sec + remainder of 2nd injection balance.)

i. $T + 525^d$: Complete Pre-re-entry ECM checkout. Prepare ECM for re-entry (transfer stores, etc. to achieve re-entry weight). Terminate experiments and load Earth return data (photos, etc.) into ECM. $T = 527^d$: Separate ECM/ESM from EMM and perform re-entry trim maneuver. Separate ECM, re-enter, and deploy parachute. Land.

Conclusions

Early manned planetary flight is presently aimed at Mars and Venus; however, among the lesser bodies of the solar family, certain interplanetary targets exist for which significantly lower mission energies are required, and from which fundamental data important to the sciences, to the technologies, and to the general fund of spaceflight experience, can be drawn. Outstanding among these more accessible lesser bodies is the well known asteroid Eros, the largest of the known close approach group and passing within 14,000,000 miles of the Earth in late January 1975.

Preliminary studies have indicated the possibility of a free return manned flight to this body using a single uprated Saturn V launch vehicle and the systems and techniques of Apollo and its contemporaries. As shown in this paper, such a mission can be conservatively implemented by the injection of a 330,000 pound Saturn V payload into a low Earth orbit. While the magnitude of this payload may appear optimistic, uprating studies place it well within the capability of thrust augmented boosters uprated by techniques already demonstrated with vehicles such as the Titan IIIC; and also within the capability of advanced versions of Saturn V. Certain questions still remain to be answered, however, such as those relating to zero g, semi-closed life support system availability, crew performance, and equipment lifetime; but answers to such questions are expected in the immediate future from programs already being implemented -- and no unsolvable problems are foreseen.

Within the limits of the study assumptions, a manned flyby of the minor planet Eros at the time of its 1975 close approach is concluded to be technically feasible, and a useful complement to the manned interplanetary program. While a cost analysis was not performed as a part of this study, the overall cost should be well below that of other manned planetary flights by an order of magnitude or less due to the extensive use of developments from other programs and the low mission energy requirements. Technically and economically Eros is one of the most easily reached objects in the Solar System.

Why go to Eros? Locked in the lesser members of the Solar family are questions of fundamental importance to the astrosciences, to the space technologies, and to the utilization of extraterrestrial resources. These questions can be answered best by direct examination, an examination that results not only in the opportunity for a better understanding of the physical universe, but also in acquiring an invaluable store of experience for the more ambitious flights to follow.

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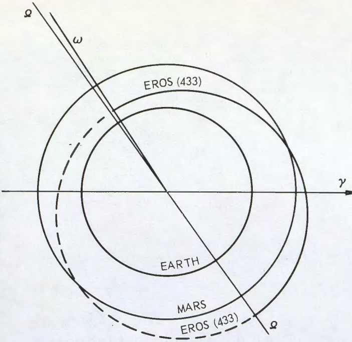
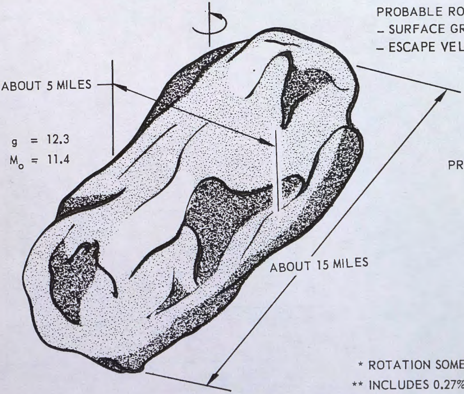


t_o (EPOCH)	JAN. 18, 1931
M_o; MEAN ANOMALY AT t_o	0.586°
ω; LONG. OF PERIHELION	177.930°
Ω; LONG. OF ASCENDING NODE	304.071°
i; ORBIT PLANE INCLINATION	10.831°
e; ECCENTRICITY OF ORBIT	0.22289
ϕ; $\sin^{-1} e$	12.879°
n; DAILY MEAN MOTION	2015.293 SEC.
a; SEMI-MAJOR AXIS	1.4581 A.U.
R_p; PERIHELION DISTANCE	1.1331 A.U.
R_A; APHELION DISTANCE	1.7831 A.U.
CLOSE APPROACH TO EARTH	0.15 A.U. IN 1975
P; PERIOD OF REVOLUTION	1.761 YEARS

FIGURE 1 ORBITAL CHARACTERISTICS OF EROS (433)

PROBABLE ROTATION = $5^h\ 16^m$ DIRECT *
– SURFACE GRAVITY ACCEL: .015 – .024 FT/SEC^2
– ESCAPE VELOCITY: 21 – 27 FT/SEC



PROBABLE COMPOSITION

SiO_2	– 38%
MgO	– 24%
FeO	– 12%
Al_2O_3	– 3%
CaO	– 2%
OTHERS**	– 21%

* ROTATION SOMETIMES GIVEN AS $2^h\ 38^m$ DIRECT
** INCLUDES 0.27% WATER; ≈ 10^{12} LB

FIGURE 2 EROS (433) – DESCRIPTION

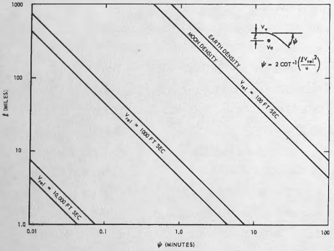


FIGURE 3 COURSE DEFLECTION FOR EROS FLYBY

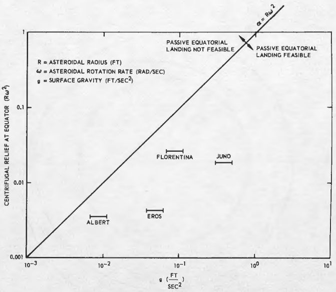


FIGURE 4 ASTEROID SURFACE GRAVITATION AND CENTRIFUGAL RELIEF

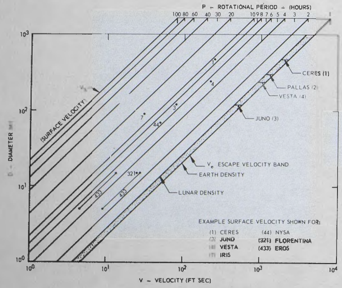


FIGURE 5 ASTEROID SURFACE AND ESCAPE VELOCITY

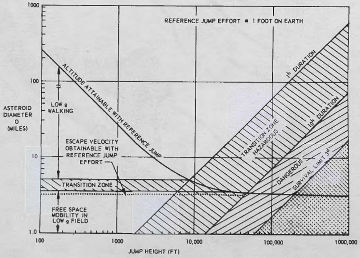
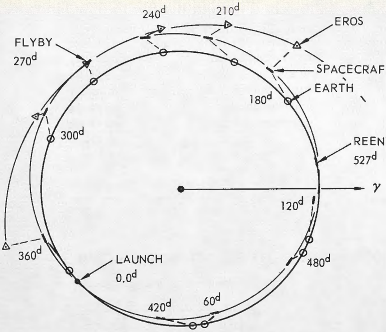
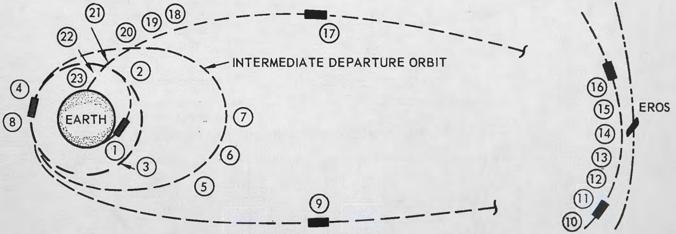


FIGURE 6 ASTEROID OPERATIONAL ZONES



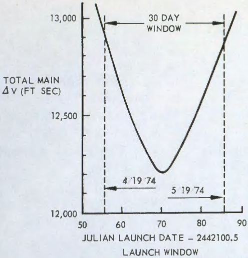
- LAUNCH 3 MAY, 1974
- FLYBY 28 JAN., 1975
- REENTRY 12 OCT., 1975
- TRANSFER ELLIPSE:
  - INCLINATION TO ECLIPTIC, 0.289°
  - SEMI-MAJOR AXIS, 1.0337 AU
  - ECCENTRICITY, 0.096655
- VELOCITIES:
  - INJECTION ΔV, 12141 FT/SEC
  - V_{rel} AT EROS, 22775 FT/SEC
  - REENTRY VELOCITY, $<$38000 FT/SEC

FIGURE 7 NOMINAL FLIGHT PATH, MODEL MISSION



1 LAUNCH
2 PARKING ORBIT
3 CHECKOUT
4 1ST IMPULSE
5 REPOSITIONING
6 JETTISON S IV B
7 ERECTION AND CHECKOUT
8 2ND IMPULSE OR ABORT
9 MIDCOURSE
10 BEGIN ENCOUNTER TRACKING
11 ENCOUNTER MANEUVER
12 ENCOUNTER ORIENTATION
13 LAUNCH PROBE
14 ENCOUNTER
15 END ENCOUNTER TRACK
16 RETURN MANEUVER
17 RETURN MIDCOURSE
18 REENTRY CHECKOUT
19 SEPARATE EMM
20 REENTRY MANEUVER
21 SEPARATE ESM
22 RE-ENTER
23 LAND

FIGURE 8 MISSION PROFILE, MODEL MISSION



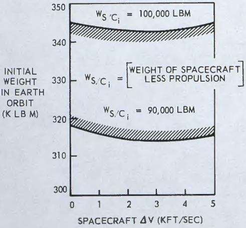
VELOCITY BUDGET

TOTAL MAIN ΔV (INJECTION PLUS RETURN) FROM A 100 N MI ORBIT	12,900 FT SEC
OUTBOUND MIDCOURSE	300 FT SEC
PRE-ENCOUNTER TRIM	100 FT SEC
POST ENCOUNTER TRIM *	100 FT SEC
RETURN MIDCOURSE	300 FT/SEC
REENTRY TRIM **	100 FT/SEC
Σ	13,800 FT/SEC

* MAY BE PERFORMED AS PART OF RETURN MANEUVER

** BASED ON SPACECRAFT GROSS WEIGHT

FIGURE 9 LAUNCH WINDOW AND VELOCITY BUDGET, MODEL MISSION



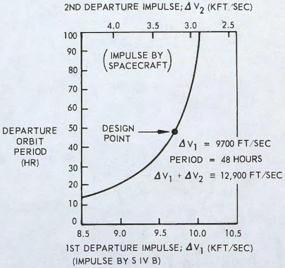
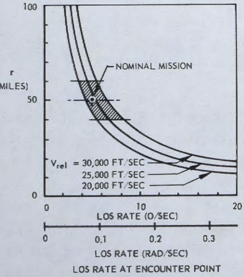


FIGURE 10 SPACECRAFT STAGING, MODEL MISSION



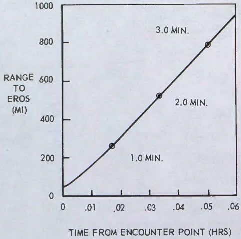


FIGURE 11 ENCOUNTER KINEMATICS, MODEL MISSION

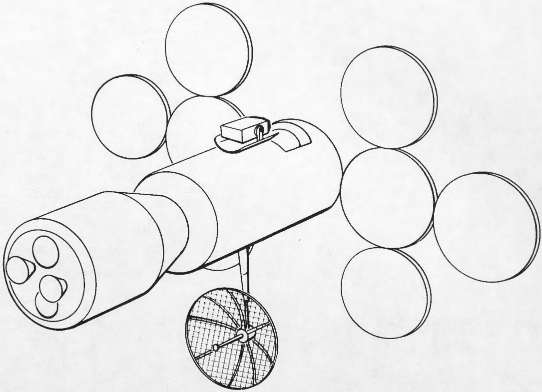


FIGURE 12 MODEL SPACECRAFT

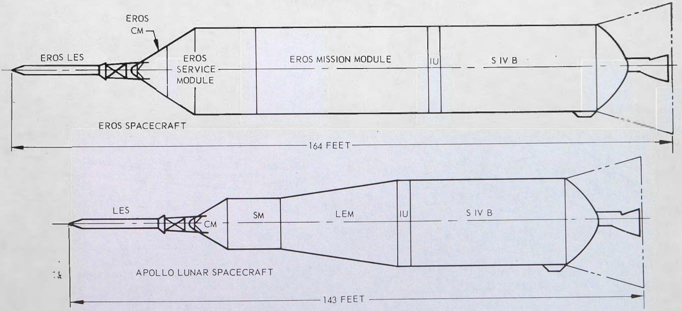


FIGURE 13 SPACECRAFT COMPARISON

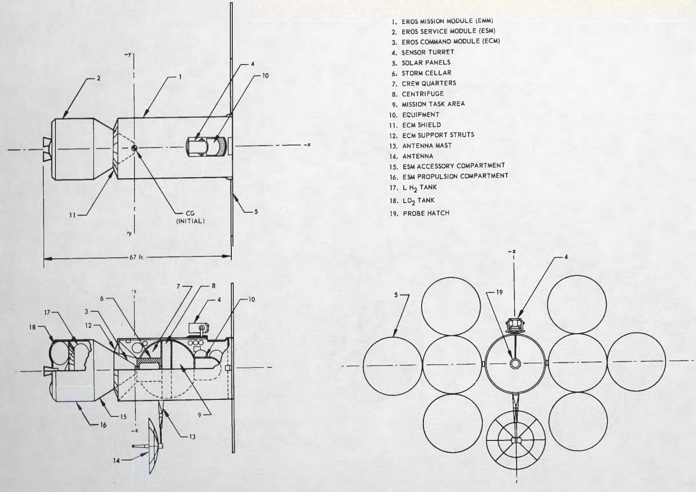


FIGURE 14 EROS SPACECRAFT (3 VIEW)

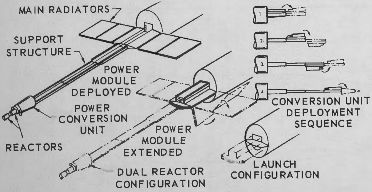


FIGURE 15 NUCLEAR POWER SYSTEM DEPLOYMENT

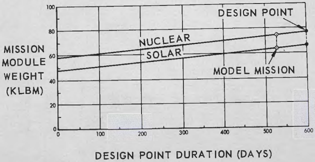


FIGURE 16 MISSION MODULE WEIGHT TRENDS

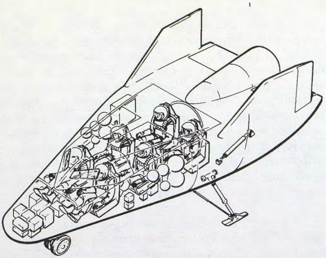


FIGURE 17 ADVANCED M-2

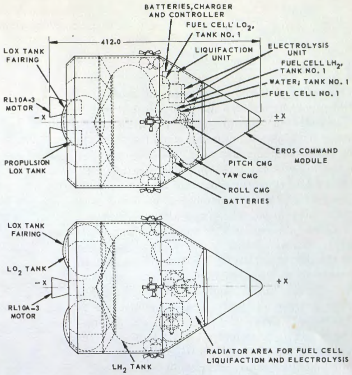


FIGURE 18 EROS COMMAND AND SERVICE MODULE

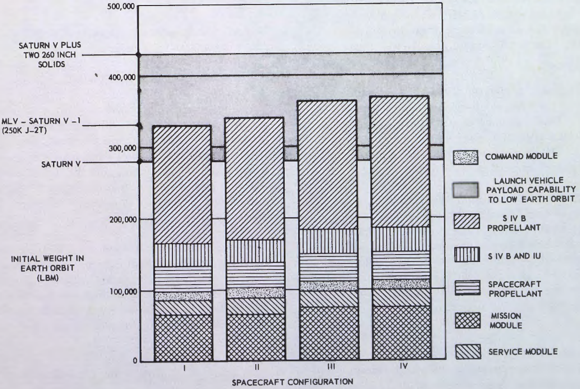


FIGURE 19 EROS VEHICLE INITIAL WEIGHT COMPARISON